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ENTRY THERMAL PROTECTION

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FOREWORD

NASA experience has indicated a need for uniform criteria for the design of space vehicles. Accordingly, criteria are being developed in the following areas of technology:

Environment

Structures

Guidance and Control

Chemical Propulsion.

Individual components of this work will be issued as separate monographs as soon as they are completed. A list of all previously issued monographs in this series can be found on the last page of this document.

These monographs are to be regarded as guides to design and not as NASA requirements, except as may be specified in formal project specifications. It is expected, however, that the criteria sections of these documents, revised as experience may indicate to be desirable, eventually will become uniform design requirements for NASA space vehicles.

This monograph was prepared under the cognizance of the Langley Research Center. The Task Manager was A. L. Braslow. The author was M. M. Sherman of Philco-Ford Corporation. A number of other individuals assisted in planning the monograph, developing the material, and in reviewing the drafts. In particular, the significant contributions made by T. Munson and I. Sacks of Avco Corporation; V. Deriugin of The Boeing Company; P. Cline of General Electric Company; L. Hearne of Lockheed Missiles & Space Company; J. W. McCown of Martin Marietta Corporation; J. E. Rogan, D. J. Chow, and R. R. Dieckmann of McDonnell Douglas Corporation; J. P. Hartnett of the University of Illinois; and R. T. Swann of NASA Langley Research Center are hereby acknowledged.

Comments concerning the technical content of these monographs will be welcomed by the National Aeronautics and Space Administration, Office of Advanced Research and Technology (Code RVA), Washington, D.C. 20546.

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ENTRY THERMAL PROTECTION

1. INTRODUCTION

The kinetic energy of a vehicle moving at high speed in a planetary atmosphere is primarily dissipated in the form of heat, most of which is rejected to the atmosphere. A fraction of this energy, dependent on the aerodynamic shape and surface properties, is transmitted to the surface of the vehicle by convection and radiation. It is the function of a thermal protection system to block, absorb, or radiate this heat and maintain both the load-carrying structure and other temperature-critical systems and components within specified temperature limits. Many high-performance vehicles will absorb such a large amount of entry heating that the required thermal protection system comprises a major portion of the total vehicle weight. Inadequate design practices can result in unnecessary weight penalties or can endanger mission success by causing failure of the vehicle structure or internal heat-sensitive equipment.

This monograph provides uniform criteria and guidance for the following steps or precautions which should be taken to ensure an adequate design of a thermal protection system:

1. Identification of the principal parameters that constrain or limit the heat-shield design – such as material properties, requirements of structure and other systems and components, geometric considerations, mission objectives, and manufacturability.

2. Identification of the important design inputs and the related disciplines interacting with thermal design.

3. Specification of the essential characteristics of the analytical design model.

4. Identification of possible critical or difficult problems that have been encountered on other vehicles.

5. Definition of the need for a supporting test program.

Launch and ascent heating, space-flight heating, gasdynamic heating during entry, aerodynamic loads, and thermal-stress analysis are planned for treatment in other design criteria monographs.

2. STATE OF THE ART

It is convenient to classify the different types of external thermal protection systems as either absorptive or radiative, depending on the primary method used to dissipate the incoming energy. The absorptive systems are further subclassified as heat sink, film and transpiration cooling, ablative, and convective. Insulation materials and internal convective cooling systems, which may be used with any of the external heat-shield concepts, are discussed separately as basic components of the thermal protection system.

Certain aspects of thermal protection systems are well understood. Radiative systems are being considered for lifting-body applications, with research efforts directed toward extending the usable service limits of candidate materials through improved surface coatings and material additives. Heat-sink systems are easy to analyze, but are used only for special applications. Ablative materials, and particularly the so-called charring ablators, are by far the most widely used thermal protection systems. This is because of their satisfactory performance, relative simplicity, and low cost, and because there has been extensive design and analytical experience with this class of materials. Considerable research is currently being conducted to gain further understanding of the mechanisms of ablation and to fashion materials to specific applications.

2.1 Radiative Systems

The radiative heat-protection system offers a simple and reliable means of thermal protection for a narrow range of operating conditions. The system is normally passive and does not involve significant mass loss. To minimize weight, the outer-surface material is usually made as thin as structural requirements permit. Energy absorption is consequently small (on the order of 2% to 5% of the incident heat flux), and the surface-energy balance can be expressed in the following equation:

$$T_W = \left(\frac{\dot{q}_S}{\sigma\epsilon}\right)^{\frac{1}{4}} \tag{1}$$

where T_W is the radiation equilibrium temperature, the maximum temperature the surface can experience for a given heat flux ($\dot{q}_S$); σ is the Stefan-Boltzmann constant; and ϵ is the surface emissivity. The radiative heat shield is limited to a maximum heat rate by its operating-temperature limit, whereas other systems are limited to a total heat input.

Once an operating-temperature limit is determined for a material, the maximum heat flux can be defined and the system can operate indefinitely at this condition. The only penalty associated with increased operating time is that more insulation is required between the heat shield and the inner load-carrying structure or payload. Because they are nearly independent of total heat input, radiative materials or panels are more appropriate for the long-duration trajectories characteristic of lifting entry vehicles than for the shorter ballistic flights.

As shown by equation (1), the maximum heating rate that can be accommodated by a radiative system increases quite rapidly with small increases in the temperature the material can tolerate. Although of lesser importance, the surface emissivity (ϵ) also affects the rate at which energy can be accommodated. Consequently, the primary development efforts on radiative systems have been directed at increasing the material service temperatures, including those of the internal insulation, and at improving high-emissivity, high-temperature coatings.

The present temperature limit for the unprotected cobalt-, nickel-, and chromium-based superalloys is about 1600° K for short-duration exposures. (Factors for conversion of U.S. customary units to SI units are presented in the appendix.) Coatings for these materials are required for an extended service life above 1400° K. The refractory metals (columbium, molybdenum, tantalum, and tungsten) retain their strength at considerably higher temperatures, but must be protected to prevent rapid material loss by oxidation at the lower temperatures. As noted in reference 1, efforts to improve oxidation resistance by alloying have had only limited success and usually result in some degradation of the mechanical properties. The molybdenum and tungsten alloys are currently not considered as practical radiative heat-shield materials because of the brittle nature of their crystalline phases.

References 2 to 4 describe some of the extensive research programs performed in recent years to develop oxidation-resistant coatings for the refractory metals. The newer slurry coatings (ref. 5) provide adequate protection for columbium to temperatures of 1550° K and for tantalum to temperatures of 1800° K to 1950° K. The service life of these coatings is temperature-dependent, and this must be accounted for if reuse is a design requirement.

The refractory ceramics (oxides, carbides, and borides) are also frequently considered for use as radiative heat shields, and under certain conditions they offer some improvement in maximum operating temperatures (ref. 6). Temperature limits of up to 2500° K are claimed for some ceramics, particularly the oxides. Some disadvantages of ceramics are their brittleness, their poor resistance to thermally induced stresses, and the fact that they are difficult and costly to fabricate and attach. A technique that has been proposed for extending the service limits of these materials consists of impregnating a porous ceramic matrix with an organic resin (ref. 7).

The maximum operating temperatures specified for the radiative materials have been obtained only on coupon samples. Their actual use at these temperatures would introduce several design problems, such as methods of attaching the panels, selection of internal insulation materials capable of operation at these temperatures, and accommodation of the thermal expansion of the panels. Even at the lower operating temperatures, thermal expansion can distort materials and cause problems (ref. 8).

Relatively large thicknesses of low-density insulation material are usually required to accomplish an effective radiative-system design (Sec. 2.5). Some design studies have shown that the use of an internal active cooling system (Sec. 2.6) can result in a thinner, lighter-weight radiative thermal protection system (ref. 9).

A principal advantage of radiative thermal protection systems is that they may eliminate the need for complete refurbishment between flights. This must be balanced against the added design complications and potential weight increase imposed by the system for attaching and supporting the heat shield and by the interface with an adjacent system of another type, usually ablative. Provision must be made for thermal expansion of the radiative panels during flight, and additional insulation is required to compensate for the heat-conduction paths provided by the heat-shield fasteners. References 10 and 11 present examples of recent efforts to design radiative thermal protection systems and their methods of attachment.

The temperature and oxidation limitations of radiative-system materials restrict their use to comparatively cool surfaces on lifting entry vehicles, or on low-performance ballistic vehicles. The only flight experience with radiative systems at high temperatures was obtained in the ASSET program (ref. 12). As a result of the limited experience available, radiative heat-protection systems are currently considered suitable for use only at operating temperatures below about 1400° K.

2.2 Heat-Sink Systems

The simplest type of absorptive thermal protection system is the heat-sink system. This was used on the noses of most of the early-generation IRBM and ICBM entry vehicles and on the afterbodies of the Mercury and Gemini vehicles. By means of a temperature rise of the external material, the heat sink absorbs aerodynamic heat without melting, vaporization, or chemical reaction. The temperature limits of most of the practical heat-sink materials are too low to permit emission of significant amounts of thermal radiation.

The principal advantage of heat-sink systems is their inherent simplicity and reliability. In addition, the thermal properties of most materials are well characterized, and the design can be accomplished by straightforward one- and two-dimensional heat-transfer calculations. These advantages are offset by the high weight of most heat-sink designs. Because of this disadvantage, heat-sink systems may not be used in the future. If they are used, they will probably be restricted to low-ballistic-coefficient vehicles and to comparatively cool areas of lifting entry bodies where there is a low total heat input.

2.3 Transpiration and Film Cooling Systems

Thermal protection systems in which liquid or gaseous material is injected into the boundary layer are classified here as transpiration and film cooling systems. In the former, the material is injected through a porous inert matrix; in the latter, the material is injected through a series of discrete slots. The injectant may or may not be chemically inert in the presence of the boundary-layer gases. The surface heat transfer is reduced in proportion to the mass-injection rate by cooling and thickening the boundary layer in such a way that the velocity and temperature gradients adjacent to the wall are greatly diminished. However, injection into a laminar boundary layer may destabilize the flow and cause premature transition to turbulent flow, with an associated increase in heating rates. Injected gases can also affect the heat-transfer and ablation characteristics of downstream surfaces (ref. 13), and this effect is normally included in the analyses.

Numerous studies have been conducted in transpiration and film cooling in a laminar boundary layer (e.g., ref. 14), and exact solutions have been experimentally verified. The transpired turbulent boundary layer has not been described analytically, but many experiments have been performed (ref. 15) and reasonably satisfactory semiempirical heat-transfer models have been formulated. Because of their empirical foundation, these models are necessarily based on data that do not cover the entire range of flight Mach and Reynolds numbers. Thus, additional work is required to improve the analytical model and to extend the range of data for the transpired turbulent boundary layer.

For a given mass-transfer rate, the heat-flux reduction is usually inversely proportional to the molecular weight of the injected gas. Therefore, for cases in which chemical reactions can be neglected, hydrogen appears to be the most efficient injectant. Liquid injectants provide the latent heat of vaporization as another means of energy dissipation, and the most promising liquid coolant is water. Other possible injectants are ammonia and lithium hydride. However, there are design problems because of the large volume change that accompanies vaporization of a liquid inside the porous surface material. There are also problems with blockage of the surface pores by solid contaminants in the liquid, although filters can be used to reduce the amount of pore blockage. Gaseous coolants tend to overcome both these difficulties, but can cause other problems because of the need for large-volume or high-pressure storage containers.

All transpiration and film systems have problems of optimum distribution of the coolant, because the fluid tends to flow to regions of lower pressure and the highest heating rates usually occur in the areas of higher pressure. Transpiration and film systems also impose additional complexities of sensing, control, and distribution devices, with an attendant loss in system reliability. Porous surfaces used for transpiration cooling are not as strong as solid surfaces and are subject to local pore blockage that can create hot spots and possibly cause the heat shield to burn through. The use of film cooling (slot injection) results in a better structure that is less sensitive to blockage, but it provides a less uniform surface temperature because of the nonuniform coolant distribution. It has been found also that slot injection is not as efficient as porous-wall injection for absorbing heat from the structure while the fluid is being injected.

Transpiration and film cooling systems are still in an early stage of development and so probably will not be used as main-body thermal protection systems in the near future. The most promising near-term applications for such systems are in areas subject to extremely high heating rates where shape changes cannot be tolerated. Transpiration and film cooling also can be beneficial where low vehicle observables or low signal attenuation is required. Reference 16 presents a feasibility study of these systems.

2.4 Ablative Systems

Most thermal protection systems designed and flown to date have been ablative-type heat shields, because of their light weight, high efficiency, and inherent simplicity and reliability. Most of these systems combine the performance of a high-temperature radiative heat shield with the heat blockage of a transpiration or film cooling system, while absorbing large amounts of energy in various phase-change processes. Because the ablation process is self-initiating and self-regulating, it also eliminates the need for sensing, control, and distribution systems.

For this discussion, ablative materials are categorized as subliming, oxidizing, melting-vaporizing, and charring.

2.4.1 Subliming and Oxidizing Ablators

The subliming ablators, typified by Teflon, decompose directly from the solid to the gaseous state; that is, the material absorbs sensible energy until the surface reaches the sublimation temperature, which is primarily a function of the local pressure. The temperature of the ablator during sublimation is also dependent on the ablation rate. As described in Section 2.3, energy is absorbed in the phase-change process and the heat flux is reduced by the transpiration effect of the evolving gases. In a low-temperature ablator, such as Teflon, the principal mechanisms of heat dissipation are transpiration cooling and the heat of depolymerization.

Teflon heat shields have been used on several ballistic-missile entry vehicles and on a few current research vehicles. Teflon is a low-to-moderate temperature ablator with moderate efficiency, and its thermal performance is reasonably well characterized. Because of its high ablation rates, a Teflon heat shield may change its shape considerably in long-duration heat pulses.

Graphite thermal protection systems have been studied intensively in recent years. Graphite, which sublimes at temperatures as high as 4000° K, accommodates or rejects the imposed heating through the mechanisms of sensible heat rise, oxidation, the latent heat of sublimation, and surface radiation. Transpiration cooling has little effect on the surface heat balance because of graphite's relatively low ablation rates. Because of its extremely high ablative effectiveness, graphite theoretically offers the minimum amount of ablation and shape change in areas subject to high heating rates, such as small-radius nose tips (ref. 17) and leading edges.

Pyrolytic graphite is an excellent high-temperature insulator because of the way it is formed. It is deposited in a series of layers and is therefore highly anisotropic. Its thermal conductivity in the direction normal to the deposition plane is almost two orders of magnitude lower than in the plane of the deposited layers. In the deposition plane, thermal conductivity is the same as for homogeneous graphite.

The chemical reaction of graphite in air has been described analytically (refs. 18 and 19) and verified by experiment (refs. 20 to 22). These analytical models allow for either one-carbon species (C_3) or two- (C and C_3) at the surface to undergo reaction with the boundary-layer gases. Because the experimental data were obtained at relatively low pressures (about one atmosphere), the correlations based on these experiments may not accurately predict oxidation in a high-pressure flight environment.

The principal disadvantages of graphite as a thermal protection system are its brittleness and low resistance to thermal stress, which restrict its maximum usable thickness. Further, graphite's high-temperature thermal and structural properties have not been reliably ascertained, and it is still subject to manufacturing inconsistencies. The pyrolytic graphites can delaminate between the deposited layers at high temperatures and high thermal stresses. Although this delamination might be tolerated for certain applications, it is generally undesirable because it is unpredictable and as yet uncontrollable. Pyrolytic graphite is also difficult and expensive to manufacture, particularly in shapes with small radii in relation to their thickness.

2.4.2 Melting-Vaporizing Ablators

The glassy, or melting, ablator is represented by such materials as quartz, pyrex, and fused silica. These materials melt at high heating rates and low surface-shear stresses, absorbing the sensible heat, and then vaporize, absorbing latent heat and providing transpiration heat blockage. Under these conditions, the glassy materials provide an adequate thermal protection system. However, when the surface shear is moderately high, or when there is a large pressure gradient, the liquid layer may be removed before it vaporizes, causing the ablative effectiveness to be substantially reduced. The analytical model for glassy ablators is well known (ref. 23) and has been verified by flight-test measurements (ref. 24).

The principal mechanical disadvantage of glassy ablators is their brittleness, although their thermal-stress resistance is high. The high surface temperatures attained during ablation(around 3000° K) do not always result in proportionately high radiant cooling because of the material's low surface emissivity. Also, since the glassy materials are transparent, self-heating by radiation to the interior can be high. Both of these difficulties can be alleviated with the use of additives in the base material to increase both emissivity and opacity.

In current vehicles, the glassy ablators are usually used only in specialized applications requiring high-temperature optical or dielectric properties, as for experiment or antenna windows.

2.4.3 Charring Ablators

The thermal protection systems of most interest are charring-ablator heat shields. These can be made of a homogeneous thermosetting resin, such as the phenolics, epoxies, or silicones; of the same resin with an organic powder, such as nylon; or of a refractory fiber, such as glass, asbestos, or graphite.

Ablative effectiveness is usually proportional to the material density, while that of insulation is inversely proportional to the density. It is therefore useful to reduce the density and thermal conductivity of most ablative materials by adding microballoons – tiny hollow spheres approximately 40 microns in diameter – made of phenolic resin or glass with a wall thickness of 1 to 2 microns (ref. 25). The additives can be so graded that the density varies uniformly through the material, and the weight is reduced with a minimal effect on the ablation performance. Some loss of char strength ordinarily accompanies the use of microballoons.

2.4.3.1 Thermosetting Resins

When an organic resin is heated, the temperature increases until the surface reaches a temperature at which the material begins to decompose (pyrolyze) and release gaseous products, leaving a porous, carbonaceous residue. The pyrolysis temperature is a function of the local pressure and ablation rate, and is relatively low, from 500° K to 800° K. As the heating continues, the pyrolysis zone proceeds into the material and the decomposition occurs below the surface. The gaseous products diffuse through the porous char to the surface, absorbing energy from the char while undergoing further decomposition (cracking). They finally exit into the boundary layer, where they act as a transpirant, and may undergo additional chemical reaction with the boundary-layer gas.

The char is primarily carbonaceous and continues to absorb heat until it reaches the temperature at which it oxidizes or sublimes (as previously described for graphite materials), or until it is mechanically removed. For lifting and moderate ballistic entries, oxidation is the dominant thermochemical char-removal mechanism. At surface temperatures below 1100° K, oxidation is limited by reaction-rate kinetics. In this regime, surface recession can be reduced appreciably by incorporating oxidation-resistant additives, such as silica. As the surface temperature increases, the oxidation rate increases exponentially until the oxygen at the surface begins to be depleted. At still higher temperatures, the surface recession is limited by the rate at which oxygen can diffuse through the boundary layer. In this regime, the mass rate of char oxidation is virtually independent of material properties. At temperatures of about 3600° K, the char sublimes.

A thick char provides an insulation barrier, radiates a large amount of heat from the surface, and is a quite effective ablator. However, the char formed on a homogeneous plastic is usually weak and brittle; thus the material is susceptible to rapid removal by mechanical shear and by spallation (possibly resulting from thermal stresses and the buildup of internal gas pressure). This reduces the insulation effectiveness of the char, exposes the cooler internal material to the surface, and results in less radiative cooling. A discussion of mechanical effects on char layers is presented in reference 26.

To improve the char-retention characteristics of the ablative resins, reinforcing fibers are usually added to the virgin material. Depending on the operating environment, these can be either organic or inorganic fibers. The fibers add strength to the char until they reach their own melting or decomposition temperature. Fiber reinforcements and other additives have also provided flexibility in the fabrication of ablative materials to specific applications. However, the use of reinforcements also increases the complexity of the analytical charring ablation model because the ablation kinetics of the fibers must be superimposed on that of the resin.

Because the fibers usually possess a higher thermal conductivity than the resin binder, fibers that are normal to the surface will increase the overall conductivity of the composite. When the fibers are placed parallel to the surface, the conductivity approaches that of the resin, but the char shear strength is greatly reduced and the material is subject to delamination. Since any variation between these extremes is possible, the fiber orientation can be selected on the basis of the particular shear-stress and heat-conduction requirements.

2.4.3.2 Elastomeric Materials

For some applications, silicone elastomeric ablators (ref. 27) have several advantages over other charring materials. They form a siliceous char layer that is essentially inert and does not recede at temperatures below approximately 1950° K. This results in extremely high ablative effectiveness in the long-duration, low-heat-flux environment characteristic of lifting entry bodies.

Elastomeric materials are frequently fabricated in a fiberglass honeycomb matrix to reinforce the char and to inhibit the flow of a melt layer which may be formed under some flight conditions. At surface temperatures above approximately 1950° K, surface recession begins, caused by melting, oxidation, and/or internal char reactions. At higher heating rates, and hence higher surface temperatures, elastomers usually provide less efficient thermal protection than that obtainable from other low-density ablators.

Many additives, including microballoons and reinforcement materials, are being used in attempts to improve the insulative and ablative efficiency of the silicone-base materials (ref. 27). Silicone elastomeric heat shields have performed satisfactorily in an entry environment on the Project Gemini and PRIME vehicles (refs. 28 and 29).

2.4.3.3 Ablation Analysis

In recent years, a large body of literature has been compiled on the theoretical and experimental performance of charring ablators. Theoretical models are all necessarily idealized and vary considerably in their complexity and ability to account for the many types of energy-absorbing mechanisms involved in the ablation process. Some analytical models designed for general classes of materials are described in references 30 to 32, while programs designed for the analysis of specific classes of materials are presented in references 33 (phenolic nylon) and 34 (phenolic silica).

Several different methods are used to describe the surface-energy balance and the subsurface pyrolysis of these analytical models. Surface recession is usually computed by (1) various types of surface combustion models; (2) empirical relations that are a function of temperature, pressure, or heat flux; or (3) empirical char-thickness limits. The pyrolysis zone is treated as either a plane layer, with a step change in density from virgin to fully charred material, or as a region of finite thickness with a variable density. The rate of pyrolysis is governed by decomposition laws of the Arrhenius type or by empirically determined functions of temperature.

Most ablation models use a "cold-wall" heating rate as one boundary condition and modify this rate to account for the actual surface temperature and the injection of gaseous ablation products into the boundary layer. These gases may react chemically with the boundary layer and will, in any event, alter the chemical composition and other characteristics of the local flow field. The properties and composition of the ablation gases are not known with any certainty. Consequently, simplified semiempirical correlations (similar to those described in Sec. 2.3) are usually employed to account for the effects of gaseous injectants by reducing the local skin friction and corresponding heat-transfer rate in the analysis. These conditions are based on experiments in which air and other gases are injected into air boundary layers. The correlation equations may also contain molecular-weight or specific-heat correction terms for use when estimates can be made of these parameters.

Further, the ablation process itself may affect the characteristics of the vehicle by changing its aerodynamic shape and stability; these changes, in turn, would change the environment and subsequent heat transfer. It is not now practical to account properly for these various complex interactions in design calculations.

A comprehensive review of the entire field of ablation technology is given in reference 35, and a discussion of the interaction between boundary layers and ablation products in reference 36.

2.4.3.4 Material Properties

The difficulties in acquiring accurate property data for materials at high temperatures are a severe handicap in the design of ablative thermal protection systems. These data, needed to verify analytical models as well as to design the thermal protection system, include both the thermophysical and thermochemical properties (enthalpy, specific heat, thermal conductivity, thermogravimetric data, etc.) and the ablation performance in a simulated entry environment. Some data accurately describing the pyrolysis kinetics of specific materials have recently become available (e.g., refs. 37 and 38), but they were obtained in the laboratory at heating rates significantly lower than those that might be encountered in flight.

High-temperature, thermophysical-property data for ablative materials and their chars have been scarce until recent years. Studies such as that reported in reference 39 have been initiated recently to provide more data of this type. An earlier compilation of high-temperature ablative-material properties is reported in reference 40. However, data such as those presented in references 39 and 40 were obtained by conventional steady-state laboratory techniques and are not necessarily representative of the transient characteristics encountered during high-speed flight.

As noted earlier, chars may be susceptible to rapid mechanical removal by spallation and by high aerodynamic shear and pressure forces. These effects are not well understood and there is a need for data on char mechanical properties and for the development of analytical methods of predicting mechanical char removal.

From the preceding discussion, it is obvious that much work remains to be done to determine the thermophysical and thermochemical properties of ablative materials. Because of the interdependence of the various parameters, improved analytical and experimental techniques are needed to separate and measure the individual parameters.

2.4.4 Ablation Material Testing

Ablation testing is used to verify analytical models, determine material properties, and screen and select candidate heat-shield materials. It is obviously desirable that tests used to verify preflight design be performed in an accurately simulated flight environment. Because of various facility limitations, however, complete similitude is usually impossible to attain, and tests must be run under several types of partial flight simulation. Results can then be evaluated by means of previously validated mathematical ablation models, or by overlapping the tests in such a way that a coherent composite picture of the important phenomena can be constructed (ref. 41).

The various types of ballistic and lifting entry vehicles, designed as they are for many different missions, undergo a wide range of entry environments. Two convenient parameters for illustrating these environments, which depend on flight performance rather than on geometry, are total enthalpy and stagnation point pressure. Figure 1 is a plot of stagnation pressure vs total enthalpy with overlays of typical entry-vehicle trajectories and maps of approximate simulation facility performance obtained from reference 42.

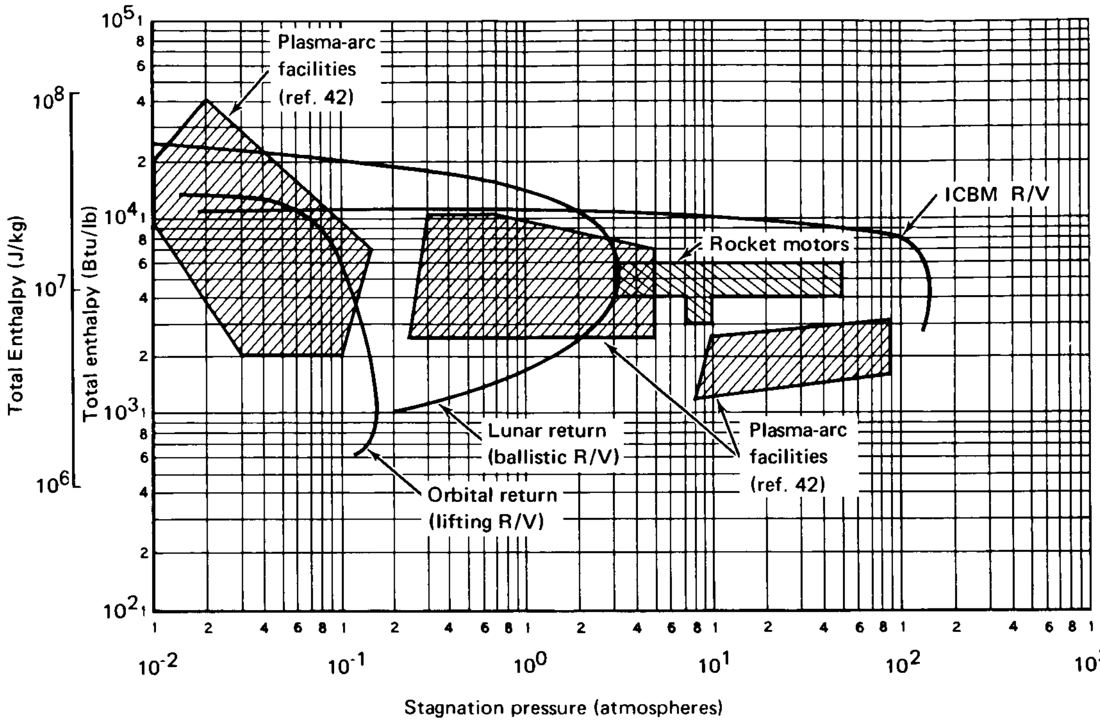

Figure 1. Comparison of flight and ablation test facility performance parameters

Two important conclusions can be immediately discerned:

1. Although large portions of some lifting-body trajectories can be closely approximated, no one facility can duplicate a complete flight environment.
2. High-pressure and high-enthalpy effects cannot be duplicated simultaneously.

For purposes of ablation simulation, slowly varying ablation is usually desired, since this condition represents the portion of major interest in most trajectories. For this reason, the facilities most applicable to entry ablation studies are the plasma-arc tunnel and the rocket-motor exhaust, both of which have operating times ranging from several seconds to continuous operation. Facilities such as the ballistic range and shock tube are of only slight interest because of their exceptionally short test times of approximately 10 msec and 100 μsec, respectively.

As indicated by figure 1, the present operating range of plasma-arc facilities covers stagnation enthalpies from 2.32×10^6 to 7×10^7 J/kg, corresponding to flight velocities of 3000 to more than 9000 m/sec, and stagnation pressures ranging from 0.001 to 15 atm; however, the maximum values cannot be achieved simultaneously. Other arc-jet facilities are available which can provide model stagnation pressures of nearly 100 atm at total enthalpies up to 6×10^6 J/kg.

Tests in the high-enthalpy facilities produce low shear stresses and, because of the low-Reynolds-number stagnation-region flow, they produce only laminar heating conditions. Turbulent shear flow, as well as laminar flow, can be achieved by attaching a pipe or shroud containing the test material to the arc exhaust. See references 41 and 43.

The rocket-motor exhaust provides extremely high stagnation-point heating rates (up to 8×10^7 W/m^2 for a nose radius of 2.54×10^{-2} m) and pressures (up to 50 atm); it can therefore be used to evaluate the effects of high shear and high pressure on ablation performance. In addition, the high heating rates are useful for the investigation of thermal-shock phenomena. The chief disadvantages of these facilities are (1) the comparatively low total enthalpies (up to 1.3×10^7 J/kg); (2) the nonuniformity of the flow field; and (3) the fact that the exhaust gas consists of propellant-combustion products with possibly uncertain composition (including discrete particles), rather than air or some other planetary atmosphere, and therefore does not duplicate the surface chemical reactions of flight. The principal advantage of rocket exhausts is that they can accommodate much larger test articles than the arc jets, and can also be adapted for pipe or shroud testing. Detailed descriptions of two rocket-motor test facilities can be found in references 44 and 45.

Because the complete flight environment cannot always be duplicated with the available test facilities, the test conditions must be carefully selected to ensure that as many critical flight conditions are represented as are practical. For most cases, however, the plasma-arc facility can provide adequate simulation up to entry velocities of 9000 m/sec. The important simulation parameters for most ablation-performance and screening tests are the heat-flux level and duration, the enthalpy, and the chemical composition of the gas streams, as shown in reference 43. For most applications, the effects of pressure on the ablation performance must be investigated.

Since the effectiveness of glassy ablators depends on the retention of the liquid melt layer, it is important that the aerodynamic-shear levels of flight be duplicated. The ablation performance of charring ablators is known to be reduced under extremely high pressures or pressure gradients which tend to remove the char layer mechanically. For vehicles that will experience unusually high local pressures, such as low-drag ballistic entry vehicles, it may therefore be necessary to simulate the pressure and shear levels, as well as the net heat flux (or heat-transfer coefficient) and enthalpy. However, since this amounts to full-scale flight duplication, which is impossible to obtain in ground test facilities, these conditions would have to be simulated in separate tests and the results applied with much caution to the flight environments.

At the exceptionally high entry velocities that will be experienced in interplanetary return missions, the radiant heating from the shock-heated air can be equal to or significantly greater than the convective heating. Some plasma-arc facilities are available (e.g., ref. 46) that can superimpose radiant heating on the convective heating, and these must be employed for performance evaluation of materials to be used in this type of environment. Testing of this type is complicated by the fact that the test models and their associated shock layers will be much smaller than those actually experienced in flight; they therefore cannot duplicate the absorption effects of the injected gases (ref. 47). Few of these tests have been conducted to date.

2.5 Insulation Materials

On ballistic entry vehicles with short, high-intensity heat pulses, the ablative heat shield usually doubles as the insulation material. Because of their much longer flight times, lifting entry vehicles must use more efficient insulators to keep the weight of the thermal protection system within reasonable limits. Ideally, the insulation material should be able to sustain and transmit the aerodynamic loads to the internal structure. However, the strong, high-density structural insulations are less effective insulators than the lower-density nonstructural materials. Most lifting vehicle designs therefore use the low-density types, which must withstand only vibration and acceleration loads while supporting their own weight. Consequently, the supports required for transmitting the loads from the outer surface to the substructure create a path for heat conduction to the interior, and tradeoffs are then required to optimize the insulation and mechanical attachment systems. An additional complication arises if the thermal protection system must be refurbished between flights. References 48 and 49 discuss the latest ablative heat-shield attachment and refurbishment techniques.

The optimum insulation material for a given application depends on the required operating temperature range. If the temperature differential and level are high, then internal radiation between the insulation particles is the dominant mode of heat transfer across the insulation. One method of reducing the radiant-heat transfer is to decrease the pore size and add a fine, particulate, opaque filler material (ref. 50). Because the gas volume is also lowered, this decreases the internal heat transfer by gas conduction; but it increases the rate of heat conduction through the solid material, as well as the material density and weight.

Some work remains to be done in the development of efficient insulation materials for entry vehicles. Efforts are directed mainly toward improving the high-temperature operating limits and the strength of the low-density insulation without increasing its thermal conductivity. Table I presents a summary of properties of some typical high-temperature insulation materials.

Table I
Properties of Typical High-Temperature Insulations

Insulation class	Temperature limit (°K)	Density (kg/m^3)	Thermal conductivity (W/m-°K)
Fibrous			
Aluminum silicate	1500	96	0.07 to 0.32
Silica fibers	1370	48 to 256	0.06 to 0.17
Zirconia fibers	1900	192	0.06 to 0.29
Potassium titanate	1480	54 to 1145	0.04 to 0.10
Glass fibers	920	8 to 242	0.003 to 0.04
Alumina fibers	2020	96	0.07 to 0.32
Ceramic foams			
Silicon carbide	2200	320	0.82
Alumina	2090	512	0.61
Zirconia	2480	736	0.14
Silica	1900	320	0.16

2.6 Convective Cooling Systems

As entry flight trajectories become increasingly long, the weight and thickness of even the most efficient thermal insulations will become prohibitively large. In these conditions it may be advantageous to remove some portion of the heat at the inner surface of the insulation by means of a convective cooling system. The convective system employs internal circulation of a fluid that absorbs heat by temperature rise and often by phase change. If it is a gas, the coolant is then dumped overboard. Because a large volume change in the cooling passages would cause difficult design problems, a change of phase is often accomplished by means of a secondary fluid. In the two-fluid system, the "primary" fluid is recirculated and the "secondary" fluid is dumped overboard after being vaporized.

As a primary thermal protection system, or even as an insulation-system replacement for moderate flight times, the convective cooling system is usually not competitive from a weight standpoint. For long flight times, however, it offers an effective supplement to thermal insulation. In addition, an onboard convective cooling system may be required on manned lifting vehicles to dissipate the thermal energy stored in the heat shield that will "soak" into the structure after a landing.

Reference 9 includes a tradeoff study in which convective cooling systems are evaluated as supplements to insulation materials in a radiation-cooled thermal protection system. To date, convective cooling systems have not been used as part of a main-body thermal protection system, but they have been utilized for the temperature control of internal components in flight vehicles.

3. CRITERIA

3.1 General

The thermal protection system of an entry vehicle shall be designed to maintain the structure at a temperature consistent with its deflection constraints and load-carrying requirements under all anticipated flight conditions.

3.2 Guides for Compliance

3.2.1 Design Constraints

The design of the thermal protection system shall, as a minimum, account for the following requirements and constraints, as applicable:

- Structural limitations.
- Compatibility with the structure.
- Thermal limitations of other systems and components (if different from those of the structure).
- Necessity for shape retention of specific areas (such as nose tips or control surfaces).
- Refurbishment or reuse.
- Manufacturability.
- Effects of sterilization.
- Effects of operation for extended periods in a space environment.
- Effects of prelaunch, ascent flight, and separation loads, including possible rocket-exhaust impingement.
- Effects of ground environment, handling, and operation.
- Communication requirements.

3.2.2 Design Inputs

At the important vehicle locations, the design of the thermal protection system shall, as a minimum, account for the maximums of:

- Heat-transfer rate.
- Duration of heat pulse.
- Total heat input (integrated with respect to time).
- Local pressure.
- Aerodynamically induced shear.

3.2.3 Design Calculations

The analytical model of the thermal protection system shall, as a minimum, have the following characteristics:

- Ability to identify and account for significant interactions between the thermal protection system and such external environments as transpiration cooling (reduction of incoming heat), chemical reactions between the external and transpired gases, and chemical reactions between the external gases and the vehicle surface.
- For ablative materials, the capability to include specific consideration of the effects of pyrolysis (charring and subsequent reactions) and surface recession (including an adequate means of correlation for high-pressure and shear effects).
- Capability for inclusion of material properties dependent on temperature or on pressure, or on both, for all possible physical states (solid, liquid, gas) of the material.
- Capability for consideration of two- and three-dimensional heat-conduction effects where they can influence temperature distributions (e.g., in radiative and heat-sink attachment systems).

An analysis of the uncertainties involved in the design process shall be performed to determine a design factor appropriate to the selected vehicle's thermal protection system.

3.2.4 Special Considerations

The design process shall identify for resolution potentially critical problems associated with geometries which are not amenable to standard design techniques, such as the following:

- Flares or deflected control surfaces.
- Control-surface actuation shafts.
- Heat-shield joints and gaps.
- Protuberances.
- Nose-tip or control-surface leading edges.
- Antenna windows.
- Cockpit canopies and optical surfaces.

3.2.5 Tests

Test data shall be used to validate analytical models and to define material properties.

Materials used in the thermal protection system shall be shown by test data to have the thermal properties and other characteristics assumed in the design calculations.

4. RECOMMENDED PRACTICES

The preferred design approach is first to identify the thermal protection requirements for all expected flight environments and establish interactions with other technical disciplines, and then to specify the essential characteristics of the analytical model, review the design for possible critical or difficult problem areas, and define the need for supporting test programs.

4.1 Design Constraints

While fulfilling its basic function (Sec. 3.1), a thermal protection system must also satisfy other structural requirements, as well as requirements of other temperature-critical systems and components. The structural- and thermal-design analyses, therefore, must be closely coordinated throughout all phases of design.

Consideration must be given to the allowable temperature levels and temperature gradients existing during all periods of significant aerodynamic loading. All candidate heat-protection systems must be examined for interrelationships with the substructure, such as matching of thermal-expansion properties; for compatibility with bond materials, if used; for bending, buckling, and expansion requirements; and, if ablative, for char-retention characteristics.

Because of the design difficulties, early attention should be given to areas required to maintain their original shapes or to undergo only a minimal shape change during entry. These areas are usually such small-radius surfaces as nose tips and control-surface leading edges that experience the maximum heating rate and total heat input because of their shape and location. Selection of the proper thermostructural system for these locations is limited normally to materials that minimize surface recession (e.g., graphite), an actively cooled structure (transpiration), or some combination of these methods (e.g., an ablative-impregnated porous matrix).

If the vehicle is to be used for a number of missions, the heat shield must be designed for reuse or refurbishment. The anticipated number of missions and the allowable time between missions must be considered in selecting material in the initial design period so that tradeoff studies can be performed between the various heat-shield concepts and the different types of segmentation and attachment methods. Possible consequences may be the imposition of machinability, bonding, handling, or other restrictions on the candidate materials.

Regardless of the thermal protection system concept, the materials must be capable of being applied, formed, or machined into the desired shape with acceptable effort and tooling. The application and curing operations for the bond materials must be considered, along with the effect of these operations on the heat shield and structure.

If prelaunch sterilization of the vehicle has been stipulated, the effects of the heating cycle on heat-shield and bond-material properties must be determined, and heat-shield attachment points should be designed to minimize the effects of differential thermal expansion.

Extended operation in a space environment before atmospheric entry imposes several material-selection and design problems. The vehicle is exposed to solar radiation, possible meteoroid impacts, a cold-temperature soak, and a vacuum environment. To maintain the desired thermal environment within the vehicle, the thermal protection system is required also to regulate and distribute incoming solar energy. This can be accomplished by means of tailored surface coatings, attitude control, heat sinks, and heat exchangers. The space environment can also alter the chemical and mechanical properties of the heat-shield materials so that their performance in the subsequent entry-heating environment is substantially degraded. Many effects of space operation on material properties can be effectively determined by means of tests in space-simulation vacuum chambers equipped to program variable radiative heating histories.

Although ascent-heating loads are nearly always much smaller than those encountered during entry, their effects on the entry thermal protection system must be evaluated. For short-duration ballistic flights, the principal effect is that of increasing the average temperature in the heat shield at the beginning of atmospheric entry. However, certain areas of the vehicle may become sufficiently hot to experience some thermochemical degradation, and this effective loss of material must be included in the design calculations. Alternate design approaches may include the application of some extra thickness of the entry heat-shield material, the addition of a thin outer layer of low-conductivity material for ascent flight, or the use of a shroud or housing for protection against ascent heating.

Additional effects that can occur before entry, and that should be accounted for in the thermal protection system design, include ascent vibration loads, separation shocks, and the impingement of separation or control-motor exhausts on the vehicle surface. In cooperation with the other system analysts and designers, the designer of the thermal protection system should account for these problems early in the design process.

The design and performance of the thermal protection system can be affected by events occurring during the prelaunch storage, transportation, testing, and maintenance operations. During storage, the vehicle may be subjected to long-duration compressive loads that would prohibit the use of external materials with high-creep characteristics (e.g., Teflon), or would require special handling methods. During transportation, the vehicle may undergo long periods of vibration loads or, if shipped by air, a cold soak as low as 220°K, which could permanently degrade the properties of some materials or increase the possibility of damage caused by low-temperature brittleness. The possible importance of such other effects as temperature and humidity extremes can be included by examining the known environmental conditions at the various storage locations and at the launch site. The designer should evaluate these ground environments and initiate any precautions necessary to protect the thermal protection system (e.g., protective coverings, special transportation or storage supports, and conditioned storage areas).

Severe and unusual communications requirements can also impose restrictions on heat-shield design and material selection. For example, the basic design can be affected if antenna windows must be located in a specific area to minimize flow-field plasma-attenuation problems or to achieve a particular line-of-sight objective. Or sometimes plasma effects must be minimized by the use of materials of extremely high purity for the ablative heat shield and windows; this can restrict the initial material selection and impose difficult handling and fabrication requirements.

4.2 Design Inputs

The aerodynamic heat-transfer (convection plus radiation) histories should be obtained for selected vehicle locations, with the number and location of the points dependent on the complexity of the vehicle geometry, and for a range of trajectories encompassing all possible entry conditions, including an abort trajectory, when applicable. Because the various limiting conditions – heat flux, total heat input, pressure, shear stress – may be experienced in different trajectories, all of the time-dependent trajectory variations and extremes should be examined.

Because the criterion for transition between laminar and turbulent flows is a most important parameter in the thermal protection system design, its selection must be closely coordinated with the entry gasdynamic-heating analyst. For example, the injection of gases from an ablative or transpiration-cooled system into a laminar boundary layer has a destabilizing effect that can cause transition to occur sooner than for undisturbed flow. In view of the difficulties in accurately predicting boundary-layer transition, a highly conservative criterion (i.e., one resulting in all, or nearly all, turbulent flow) may be acceptable when the thermal protection system comprises a small percentage of the total vehicle weight.

4.3 Design Calculations

Because of the many satisfactory mathematical techniques available for analysis of the different types of thermal protection systems, it is not practical to recommend a specific technique for use in design calculations. Several representative techniques for analysis of transpiration and ablative protection systems are summarized in Sections 2.3 and 2.4, and attention is directed to reference 35 for a discussion of methods applicable to ablative heat shields. The mathematically simpler cases of radiative and heat-sink systems can be solved by most heat-conduction computer programs with the proper surface-energy balance. See reference 9, for example. The only requisite for any analytical design technique is previous verification by ground- and/or flight-test data correlations or comparison with other similarly proven analytical models. Use of the model outside the region of verification is, of course, subject to the same uncertainties that exist in material performance, and both the model and the materials should ultimately be verified to some degree by tests.

As in the case of structural design, some degree of conservatism must be applied in the design of a thermal protection system to account for the uncertainties that are introduced at nearly all stages of the design process. One common design technique consists of simply using a conservative set of assumptions in computing the thermal protection system requirements. These assumptions can include the use of trajectory extremes, all-turbulent flow (or very early transition), and pessimistic material properties.

A less conservative technique involves use of a design factor determined by individually computing or estimating the effects of specific parameters on the thermal protection system design, and then by determining the overall uncertainty range by means of a root-mean-square calculation. The parameters usually included are the trajectory variations, the heat-transfer inputs, the transition Reynolds number, the various material thermochemical and mechanical properties, and the analytical model. For ablative heat shields, the design factor should be applied to the computed shield thickness. Examples of the relative importance of several of these properties on the computed thickness of some ablative heat shields have been published in references 51 and 52.

For a radiative system, however, increasing the material thickness is not necessarily conservative, and it is recommended that the allowable service-temperature limit be reduced by an amount based only on the estimated uncertainty in the gasdynamic-heating calculations.

In transpiration or film cooling, conservatism is introduced by increasing the coolant flow rate. The increase in coolant flow rate should be calculated on the basis of the uncertainties in the heat flux and on the estimated effectiveness of the sensing, control, and distribution techniques.

4.4 Special Considerations

Most of the special considerations cited in the criteria refer to areas or components involving complex geometric shapes; hence, the major design problem is usually the prediction of local pressures and heating rates. This problem is nearly always solved best by the performance of heat-transfer tests involving scale models, even though the complete external environment cannot always be duplicated. Further, complex shapes can create environments that cause unpredictable material responses and interactions. The material responses, therefore, must also be closely examined in conjunction with the heat-transfer tests.

Flares and deflected control surfaces may be subject to increased heating caused by shock-wave interaction or boundary-layer separation and reattachment. Owing to their exposed position, these surfaces can also be subjected to severe mechanical erosion from impinging char or fiber particles produced by upstream ablating surfaces. Particle erosion is difficult to predict analytically and must often be evaluated with large-scale test models in a facility having a large test section. Methods for minimizing the effects of particle erosion include (1) utilization of materials on the exposed surfaces having high resistance to particle impact; (2) reduction of the inclination angle of the exposed surfaces to minimize the normal velocity component of the impinging particles; and (3) substitution of a noncharring (or otherwise low-particle-producing) heat-shield material (e.g., Teflon or graphite) in large areas of the upstream surface to eliminate the source of the particles. This last method can result in large weight penalties or additional design complexity, but may be required to attain vehicle flightworthiness.

Control-surface actuation shafts must be shielded from direct heating and erosion, and seals must be provided for protection of the shaft bearings from high-temperature gases.

Because of field breaks, access doors, or antenna and experiment windows, the thermal protection system frequently contains many joints, gaps, and protuberances. These regions often produce unpredictable heating environments and material responses. The thermal protection system design in these regions should be closely coordinated with the gasdynamic-heating analyses.

Nose tips or control-surface leading edges constructed of ablative materials will change shape during the entry trajectory and cause a corresponding change in the incident heat-transfer rate that should be included in the design calculations. Also, the effects of high pressures and pressure gradients on the performance of the ablation material should be examined. With transpiration-cooled surfaces, the effects of the injected gases on the heating of the downstream surfaces should be included.

In addition to the ablation and back-face temperature limits of antenna windows, other problems that must be considered are the thermal stresses in the window material and the effect of the window on the surrounding heat shield (edge effects and sealing and mounting requirements). Because the antenna windows probably will have different ablation- and surface-recession characteristics than the heat-shield material, the possibility of premature boundary-layer transition and protuberance-heating effects caused by the differential surface recession should be investigated. The effects of easily ionized contaminants in the window material that may impair signal transmission should also be considered along with the procurement and handling problems associated with high-purity requirements.

Some vehicles may require transparent canopies or other optical surfaces that will require protection from entry heating. A protective covering that can be jettisoned during the later, cooler portions of the trajectory should be used to protect a canopy; optical surfaces should be located so as to avoid damaging environments during entry.

4.5 Tests

Tests to obtain material properties or to verify the analytical model used in the design calculations frequently cannot be performed in a completely simulated flight environment. A series of tests should therefore be performed using the simulation parameters determined by analysis to be most important for each material of interest. Transient effects, for example, are frequently important during the period of maximum flight heating, but even test conditions duplicating the extremes of the flight environment can usually be achieved only as steady states. It is therefore often desirable to try to duplicate the predicted total heat input by various combinations of heating rates and exposure times. In this manner, the transient effects of char formation on the ablative material can be at least partially simulated. If the test models are instrumented with thermocouples and the measured temperature histories, as well as the mass loss and surface recession, are analytically matched, much insight can be gained into the material properties and the analytical model.

When the tests are performed for the specific purpose of verification of the analytical model, extreme care should be exercised in the design of the test model to eliminate such effects as side heating and heat "shorts." Care should be taken also in the selection and installation of measuring devices, and in the interpretation and analysis of test data. Successful prediction of test results under various conditions adds confidence to extrapolations from the analytical model to the actual flight environment. If the required simulation is not possible, serious consideration should be given to performance of a subscale or unmanned full-scale flight test.

Both before and during manufacture of the thermal protection system, several steps should be performed to ensure that the flight hardware will possess the same properties obtained in laboratory or model-shop fabrication and used in the design calculations. The first step is to establish firm specifications for material procurement and processing. Following this, all steps in the manufacturing and assembly process should be reviewed with quality-control and inspection personnel to ensure that inspections will be performed and accurate records maintained of all pertinent properties and measurements.

APPENDIX
CONVERSION OF U.S. CUSTOMARY UNITS TO SI UNITS

The International System of Units (SI)(ref. 53) was adopted in 1960 by the Eleventh General Conference on Weights and Measures held in Paris, France. Conversion factors required for units used in this monograph are given in the following table:

Physical quantity	U.S. customary unit	Conversion factor[a]	SI unit
Density	lbm/ft^3	16.02	kg/m^3
Enthalpy	Btu/lbm	2.32×10^3	J/kg
Heat-transfer rate	Btu/ft^2-sec	1.13×10^4	W/m^2
Length	ft	0.305	m
	in.	2.54×10^{-2}	m
Pressure	atm	1.013×10^5	N/m^2
	lbf/ft^2	47.88	N/m^2
Temperature	°F	5/9 (°F+460)	°K
Temperature rise	°F	0.556	°K
Thermal conductivity	Btu/ft-sec-°R	6.23×10^3	W/m-°K

[a]Multiply value given in U.S. customary unit by conversion factor to obtain equivalent value in SI unit.

Prefixes to indicate multiples of units are as follows:

Prefix	Multiple
micro (μ)	10^{-6}
milli (m)	10^{-3}
centi (c)	10^{-2}
kilo (k)	10^3
mega (M)	10^6

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SP-8003 (Structures)	Flutter, Buzz, and Divergence, July 1964
SP-8004 (Structures)	Panel Flutter, May 1965
SP-8005 (Environment)	Solar Electromagnetic Radiation, June 1965
SP-8006 (Structures)	Local Steady Aerodynamic Loads During Launch and Exit, May 1965
SP-8007 (Structures)	Buckling of Thin-Walled Circular Cylinders, September 1965
SP-8008 (Structures)	Prelaunch Ground Wind Loads, November 1965
SP-8009 (Structures)	Propellant Slosh Loads, August 1968
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